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RESEARCH MEMORANDUM

EXPERIMENTAL STUDY OF A METHOD OF DESIGNING THE SWEPTBACK-WING—FUSELAGE JUNCTURE TO REDUCE THE DRAG AT MODERATE SUPERSONIC SPEEDS

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EXPERIMENTAL STUDY OF A METHOD OF DESIGNING THE SWEPTBACK-WING—FUSELAGE JUNCTURE TO REDUCE THE DRAG AT MODERATE SUPERSONIC SPEEDS

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SUMMARY

An investigation has been made in the Langley transonic blowdown tunnel at Mach numbers between 0.81 and 1.42 to determine the effects on the zero-lift pressure drag of a sweptback-wing—fuselage combination due to combining a specific fuselage contour in the wing-fuselage juncture with the area distribution obtained from application of the principle of the supersonic area rule. The general configuration tested consisted of a thin 45° sweptback wing of aspect ratio 4 in combination with a body of fineness ratio 9.2. The ratio of fuselage frontal area to wing plan-form area was 0.11. Pressure-drag reductions greater than those obtained by an axisymmetrical indentation of the fuselage in accordance with the principles of the supersonic area rule as applied for a design Mach number of 1.2 were obtained at Mach numbers between 1.2 and 1.4, the highest test Mach number, with a configuration having the same longitudinal distribution of area but a localized fuselage shaping in the wing-root juncture in accordance with the curvature of the streamlines over an infinite sweptback wing.

A comparison of the zero-lift drag variation with Mach number of the wing-fuselage configuration axisymmetrically indented according to the principle of the supersonic area rule (designed for a Mach number of 1.2) with the wing-fuselage configuration axisymmetrically indented according to the principle of the transonic area rule (designed for a Mach number or 1.0) indicates that, for this configuration, the Mach number 1.0 indentation becomes inferior to the Mach number 1.2 indentation at Mach numbers above about 1.25.

INTRODUCTION

The concept of shaping the fuselage of a sweptback-wing—fuselage combination in such a way as to combine the curvature of the streamlines

over an infinite sweptback wing with the longitudinal area distribution obtained from application of the transonic area rule (ref. 1) was advanced in reference 2. Also presented in reference 2 are experimental data which indicated that this method of fuselage shaping resulted in reductions in pressure-drag coefficient greater than those obtained through the use of an axisymmetric application of the transonic area rule alone. It was then considered desirable to determine whether the method of fuselage contouring used in reference 2 could be combined with the principle of the supersonic area rule (ref. 3) to afford drag reductions greater than those offered by an axisymmetric application of the principle of the supersonic area rule alone.

The present investigation was undertaken, therefore, to obtain such experimental information on a thin sweptback-wing—fuselage configuration of high fineness ratio. Tests were also made of a fuselage modified in accordance with the principle of the transonic area rule in order to determine the drag-reducing effectiveness of an axisymmetric Mach number 1.0 fuselage indentation as compared with the axisymmetric supersonic fuselage indentation.

The investigations were made in the Langley transonic blowdown tunnel through a range of Mach number from 0.81 to 1.42 with a corresponding variation in Reynolds number from 2.5×10^6 to 2.9×10^6 based on the wing mean aerodynamic chord at an angle of 0°.

SYMBOLS

A_b area of the model base, 0.866 sq in.

C_{D_T} total-drag coefficient, $\frac{\text{Measured drag}}{q_o S}$

C_{D_b} base drag coefficient, $-(p_b - p_o) \frac{A_b}{q_o S}$

C_D net drag coefficient, $C_{D_T} - C_{D_b}$

p_b measured base pressure

p_o free-stream static pressure

M_o free-stream Mach number

q_o free-stream dynamic pressure, $0.7\ p_oM_o^2$

S total plan-form area of wing, 12.960 sq in.

MODELS

The general configuration tested consisted of a sweptback wing of aspect ratio 4, taper ratio 0.6, 45° sweep of the quarter-chord line, and NACA 65A004 airfoil sections in the stream direction mounted in the midwing position on a fuselage of fineness ratio 9.2. The ratio of fuselage frontal area to wing area was 0.11. The basic fuselage, which was an arbitrary body of revolution, was modified to obtain the three different fuselages tested as follows:

One of the fuselages tested was indented by application of the principle of the supersonic area rule (ref. 3) to the basic wing-body combination. As in reference 3, the indentation was determined by averaging three area developments of the exposed wing which were obtained by taking oblique cuts made tangent to the design Mach cone at 0°, 45°, and 90° roll angles. The 45° cut was weighted at twice the 0° and 90° cuts in obtaining the average area development for the wing. This average wing area development was subtracted from the basic fuselage area development to obtain the supersonic indentation. The indentation was made axisymmetrically and was designed for a free-stream Mach number of 1.2. The ordinates obtained for the fuselage are presented in table I and a sketch and photographs of the configuration are presented as figures 1 and 2, respectively.

A second fuselage was obtained by combining with the longitudinal area development that resulted from the application of the supersonic area rule a fuselage contour shaped in accordance with the calculated streamline shape that would exist on the wing surface at a free stream Mach number of about 1.2 if the sweptback wings were of infinite span and the fuselage were not present. For the present case, the streamline shape used was computed in the same manner as the shape used in reference 2. The sweepback angle of the 50-percent-chord line of the wing was used as the sweepback of the wing of infinite span. Inasmuch as theoretical airfoil pressure distributions have been shown by numerous investigations to agree with experimentally determined values at Mach numbers less than critical, theoretical pressure distributions were used for the present streamline derivation in the absence of suitable experimental values. The pressure distribution used was obtained from reference 4 and the corresponding velocity distribution was adjusted for first-order compressibility effects according to Prandtl-Glauert rule.

As in reference 2, the slope of the resultant velocity at each point along the chord was obtained by combining the local velocity normal to the c/2 line with the component of the free-stream tangent to the c/2 line. These slopes were multiplied by incremental distances to obtain lateral displacements which were summed progressively from leading to trailing edge of the wing. As previously mentioned, the wing-surface—fuselage intersection line was made to conform to this calculated shape. Behind the wing trailing edge, the streamline shape was arbitrarily faired into the plan-form shape of the fuselage which had been symmetrically indented according to the supersonic area rule. By use of elliptic shapes, the fuselage cross-sectional shape was distorted to allow the longitudinal area development of the fuselage indented according to the supersonic area rule to be maintained while satisfying the streamline shape in the wing root. The distortion was made only when necessary - the remainder of the cross-sections of the fuselage being circular. The design ordinates for this fuselage are presented in table II and a sketch and photographs of the configuration are presented as figures 1 and 2, respectively.

The third fuselage tested was indented according to the principle of the transonic area rule (ref. 1). This fuselage was axisymmetrically indented as was the fuselage obtained from the principle of the supersonic area rule. Design ordinates for this fuselage are presented in table III and a sketch and photographs are presented as figures 1 and 2, respectively.

A comparison of the longitudinal area distributions for the configurations tested is presented in figure 3.

APPARATUS

The investigation was made in the Langley transonic blowdown tunnel which has an octagonal slotted test section measuring 26 inches between flats. The models were mounted on an internal electrical strain-gage balance supported by a sting at an angle of attack of 0°. The angle was set with a sensitive inclinometer and was unchanged for all tests. The force data were recorded by photographing self-balancing potentiometers.

The model base pressure was measured with an open end tube inserted through the center of the sting into an open section of the balance. The pressure so measured was the average static pressure in the open annulus around the sting in the plane of the base. The base pressures as well as the pressures required for the determination of Mach number, dynamic pressure, and Reynolds number were recorded by a quick-response flight-type pressure recorder.

TESTS

The tests were made through a range of Mach number from 0.81 to 1.42 at Reynolds numbers ranging from 2.5×10^6 to 2.9×10^6 based on the wing mean aerodynamic chord. For the ratio of model to tunnel size used, reference 5 indicates negligible tunnel wall interference at subsonic speeds. At supersonic speeds, the data are also equivalent to free-air values except for a range of Mach number where wall-reflected disturbances affect the measurements. Based on the measurements of base pressure and experience with models of similar size, it appears that for the fuselage alone the results would be affected by the wall reflections between a Mach number of about 1.03 to 1.14. The reflected disturbances intersect the tip of the wing of the wing-body combination to a slightly higher Mach number. The wing-tip interference, however, should have no effect on the comparison of the drag of different wing-fuselage combinations inasmuch as it should be not only very small but the same for all configurations compared.

The tests were made with a 1/8-inch-wide strip of 0.001- to 0.002-inch-diameter carborundum particles running spanwise and located on both the upper and lower wing surfaces at 10 percent of the local wing chord behind the wing leading edge. The particles were blown on a wet strip of thinned shellac. There was also a similarly constructed 1/4-inch-wide roughness band around the forebody of the fuselages located 1 inch back from the nose. Inasmuch as the same wing was used for all of the tests, the wing roughness strip did not change between configurations. Care was taken to insure that the fuselage roughness band had the same degree of roughness for each of the three configurations.

The base drag coefficient was obtained from the difference between the measured base pressure and the free-stream static pressure and was algebraically subtracted from the measured total-drag coefficient to obtain the net drag coefficient. The drag data measured at Mach numbers greater than about 1.15 were corrected for buoyancy effects resulting from longitudinal gradients in test section Mach number. The estimated maximum errors in total-drag and base drag coefficients are ±0.001 and ±0.0005, respectively. The general level of accuracy is believed to be better than these limiting estimates.

RESULTS AND DISCUSSION

The basic drag characteristics at an angle of attack of 0° of the three configurations tested are presented in figure 4 as a function of

Mach number. Presented are the total-drag coefficient, base drag coefficient, and net drag coefficient, all based on the total wing area.

As was noted previously, the tests were made with roughness strips to fix transition on the wing and fuselages, thereby eliminating the possibility of changes in the viscous drag between the different configurations as a result of changes in the location of boundary-layer transition. The order of obtaining data points for each configuration started with the lowest test Mach number and progressed to the highest test Mach number. After the test Mach number range was covered, a check point was taken at a subsonic Mach number to insure that the drag due to the roughness particles had not changed during the tests. The check points are noted by the flagged symbols on the data plots and indicate that any change in drag due to roughness blowing off the model was well within the accuracy of the tests.

The fairing of the drag-coefficient curves through the shock reflection interference range $(1.03 < M_o < 1.14)$ is purely arbitrary and hence no valid comparison can be made of the drags in this range of Mach number.

For all of the configurations tested, the increase in drag coefficient between a Mach number of 0.85 and 1.2 is about 0.007, an indication that the general configuration was a fairly low pressure-drag form.

Comparison of the Drag Characteristics of the Streamline Contoured and Axisymmetrically Indented Fuselages

A comparison of the net drag coefficient for the configuration axisymmetrically indented according to the supersonic area rule and the combination of supersonic area rule and streamline contoured configuration is presented as figure 5. Inasmuch as the subsonic level of drag coefficient for both configurations is nearly the same, the figure also serves to show a comparison of the variation of pressure drag coefficient with Mach number for the two configurations.

Combining the streamline shape to the fuselage sides with the fuselage area development obtained from application of the supersonic area rule resulted in a reduction in pressure drag in the range of Mach number between about 1.2 and 1.4. The magnitude of the reduction for the present case amounted to approximately 0.002 in net drag coefficient or about 25 percent of the total pressure drag rise. It is not clear from the results of these tests why the maximum drag reduction occurred at a Mach number near 1.3 rather than at the design Mach number of 1.2. There are a number of factors, however, that may have bearing on this result. Among these are the growth of the boundary layer on the fuselage and wing which may cause an appreciable change in the effective shape of

the fuselage contour, the arbitrary fairing made of the fuselage shape behind the wing trailing edge, and the three-dimensional effects of the finite wing and fuselage on the pressures in the wing root which were neglected in designing the streamline shape.

It is indicated that pressure-drag reductions greater than those afforded by axisymmetrically indenting the fuselage according to the principle of the supersonic area rule can be obtained by combining with the indentation according to the supersonic area rule a localized indentation in the wing-fuselage juncture in accordance with the streamline flow over a sweptback wing of infinite span. It should be pointed out, however, that the magnitude of the pressure-drag reductions attainable by contouring the fuselage in accordance with the streamline shape would depend upon the overall slenderness of the configuration as well as on the relative size of the fuselage and wing. It is probable that the effectiveness of the streamline contouring in affording pressure-drag reductions would decrease with respect to the effectiveness of the area-rule indentation as the overall slenderness of the configuration is increased or the relative size of the fuselage is decreased.

A comparison of the variation of net drag coefficient with Mach number for the configurations whose fuselages were axisymmetrically indented according to the principles of supersonic and transonic area rule are presented in figure 6. The comparison shows the difference in drag coefficient at any Mach number within the test range that may be expected from the differences in fuselage design for a Mach number of 1.0 (transonic area rule) and a Mach number of 1.2 (supersonic area rule). It is indicated that, for the configuration tested, the configuration indented according to transonic area rule had a slightly lower zero-lift drag at Mach numbers near 1.0. At Mach numbers greater than about 1.25, the drag of the Mach number 1.0 indented body increased with Mach number such that the increase in drag coefficient above that of the Mach number 1.2 indented body was about 0.003 at a Mach number of 1.4.

CONCLUDING REMARKS

Zero-lift drag measurement at Mach numbers from 0.8 to 1.4 have been made on a thin 45° sweptback wing of aspect ratio 4.0 in combination with a fuselage of fineness ratio 9.2 to determine the change in pressure drag of the configuration due to contouring the sides of the fuselage to conform approximately to the streamline shape over a sweptback wing of infinite span. The ratio of fuselage frontal area to wing plan form area was 0.11.

For the configurations investigated, pressure-drag reductions greater than those obtained by an axisymmetrical indentation of the fuselage in accordance with the principles of the supersonic area rule were obtained at Mach numbers between 1.2 and 1.4 with a configuration having the same longitudinal distribution of area and a localized fuselage contour in the wing-root juncture shaped in accordance with the streamline flow over an infinite sweptback wing.

A comparison of the variation of zero-lift drag coefficient with Mach number for the fuselage axisymmetrically indented according to the supersonic area rule (designed for a Mach number of 1.2) with that of a fuselage axisymmetrically indented according to the transonic area rule (designed for a Mach number of 1.0) showed no large differences in drag coefficient except for Mach numbers above 1.25 where the drag of the body indented according to the transonic area rule became significantly greater.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., July 22, 1955.

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5. Wright, Ray H., and Ward, Vernon G.: NACA Transonic Wind-Tunnel Test Sections. NACA RM L8J06, 1948.

TABLE I

DESIGN COORDINATES FOR THE FUSELAGE INDENTED

ACCORDING TO THE SUPERSONIC AREA RULE

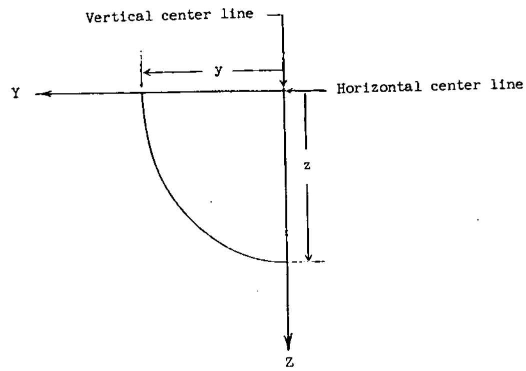
Body station, x	Body radius, r
0.000	0.000
.050	.060
.100	.090
.400	.196
.800	.278
1.200	.340
2.000	.440
2.500	.491
3.000	.538
3.500	.575
4.000	.604
4.800	.631
5.200	.638
5.600	.645
6.000	.651
6.400	.644
6.800	.632
7.200	.613
7.600	.597
8.000	.595
8.400	.593
8.800	.591
9.200	.588
10.000	.593
10.800	.568
11.600	.525

TABLE II

DESIGN COORDINATES FOR THE COMBINATION FUSELAGE

Body station, x	Body radius, r
0	0
.4	.197
.8	.279
1.2	.341
1.6	.394
2.0	.441
2.4	.483
2.8	.521
3.2	.554
3.6	.582
4.0	.604
4.4	.619
4.8	.631
5.2	.638
5.6	
6.0	
6.4	
6.8	See body cross-sectional ordinates
7.2	
7.6	
8.0	
8.4	
8.8	
9.2	
10.0	.593
10.8	.561
11.6	.520

Body cross-sectional view



Body cross-sectional ordinates

x = 5.6		x = 6.0		x = 6.4		x = 6.8		x = 7.2		x = 7.6	
z	y	z	y	z	y	z	y	z	y	z	y
0	0.650	0	0.665	0	0.650	0	0.620	0	0.590	0	0.560
.1	.642	.1	.657	.1	.642	.1	.613	.1	.583	.1	.554
.2	.617	.2	.631	.2	.617	.2	.590	.2	.562	.2	.535
.3	.574	.3	.586	.3	.574	.3	.549	.3	.525	.3	.501
.4	.507	.4	.517	.4	.506	.4	.487	.4	.468	.4	.450
.5	.405	.5	.411	.5	.404	.5	.393	.5	.382	.5	.373
.55	.331	.55	.334	.55	.329	.55	.325	.55	.322	.55	.321
.60	.224	.60	.221	.60	.221	.60	.230	.60	.239	.60	.251
.625	.135	.625	.123	.625	.131	.625	.157	.625	.179	.625	.204
.636	.063	.636	0	.638	0	.638	.097	.638	.137	.638	.173
.639	0					.646	0	.646	.103	.646	.151
								.656	0	.655	.122
										.663	.086
										.671	0

x = 8.0		x = 8.4		x = 8.8		x = 9.2	
z	y	z	y	z	y	z	y
0	0.560	0	0.565	0	0.578	0	0.594
.1	.554	.1	.558	.1	.571	.1	.586
.2	.534	.2	.538	.2	.549	.2	.563
.3	.499	.3	.502	.3	.510	.3	.522
.4	.447	.4	.447	.4	.451	.4	.461
.5	.368	.5	.365	.5	.360	.5	.356
.55	.313	.55	.307	.55	.294	.55	.282
.60	.238	.60	.227	.60	.199	.60	.166
.625	.187	.625	.169	.625	.120	.625	0
.638	.152	.638	.128	.636	.056		
.646	.126	.646	.093	.639	0		
.655	.087	.655	0				
.663	0						

TABLE III

DESIGN COORDINATES FOR THE FUSELAGE INDENTED

ACCORDING TO THE TRANSONIC AREA RULE

Body station, x	Body radius, r
0.000	0.000
.050	.060
.100	.090
.400	.196
.800	.279
1.200	.341
1.600	.394
2.000	.441
2.400	.483
2.800	.521
3.200	.554
3.600	.582
4.000	.604
4.400	.619
4.800	.631
5.200	.638
5.600	.645
6.000	.651
6.400	.644
6.800	.633
7.200	.622
7.600	.613
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8.400	.608
8.800	.608
9.200	.606
10.000	.593
10.800	.561
11.600	.520

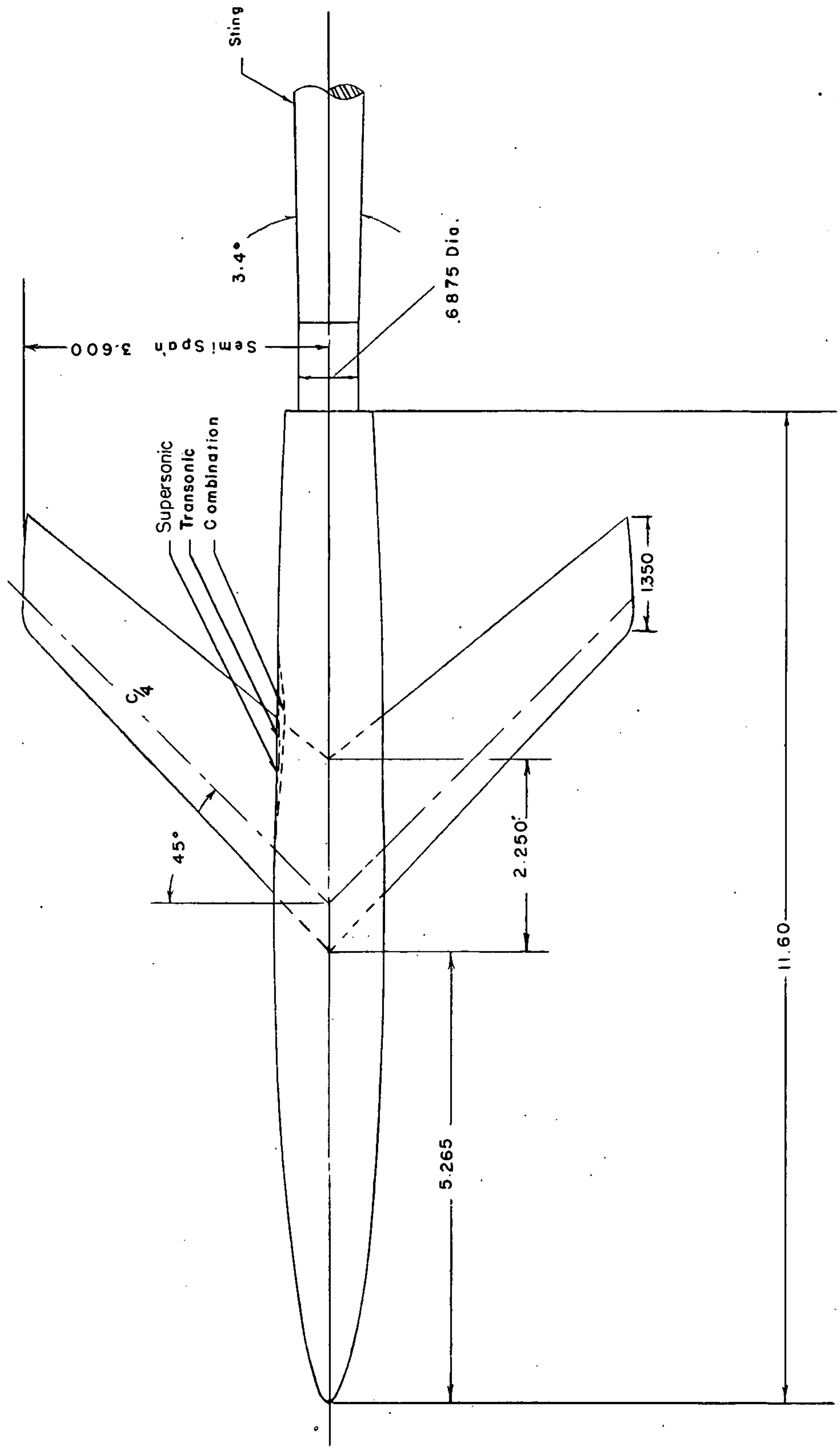
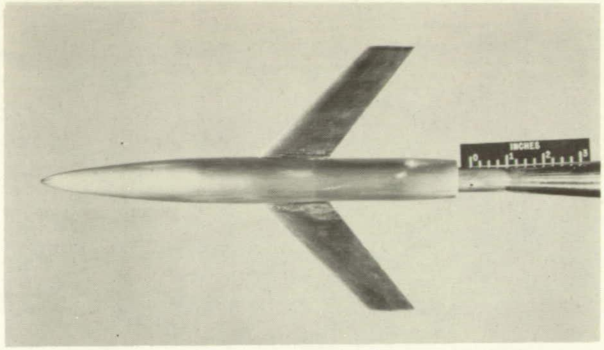
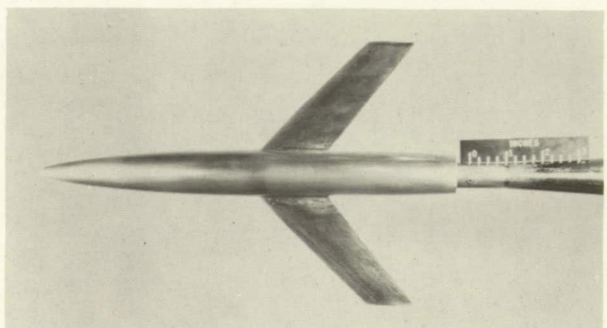


Figure 1.- Diagrammatic sketch showing dimensions of the three configurations tested. All dimensions are in inches.

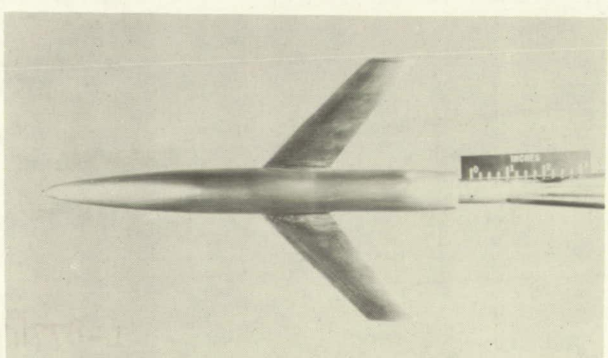


L-87742

Configuration indented according to supersonic area rule



Combination configuration L-87743

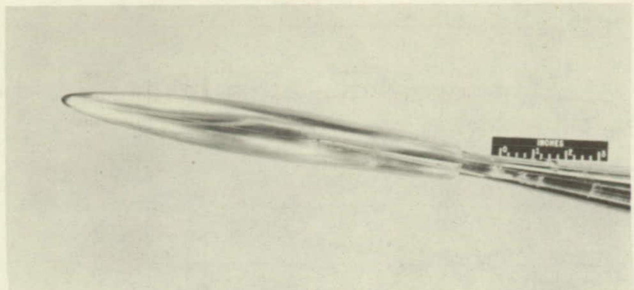


L-87744

Configuration indented according to transonic area rule

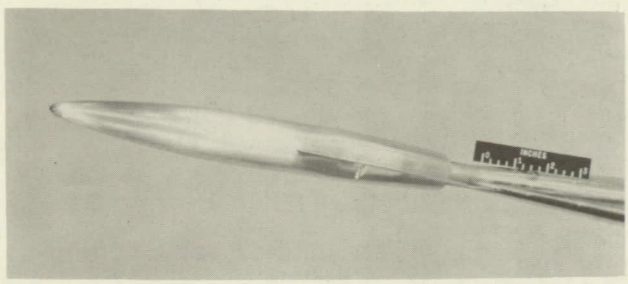
(a) Plan-view photographs.

Figure 2.- Plan view and three-quarter side view of the three wing-body configurations tested.



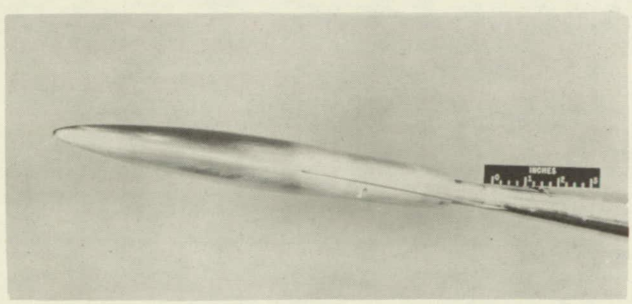
L-87746

Configuration indented according to supersonic area rule



L-87745

Combination configuration



L-87747

Configuration indented according to transonic area rule

(b) Side-view photographs.

Figure 2.- Concluded.

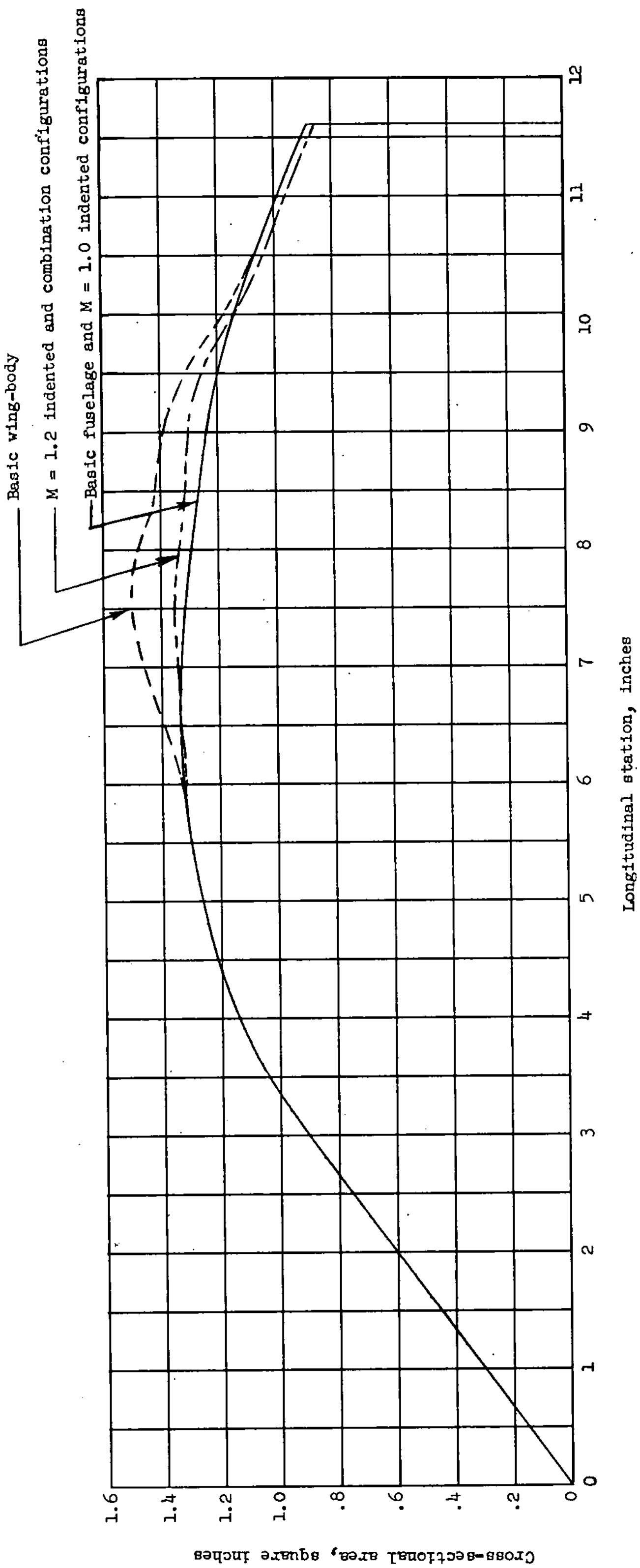
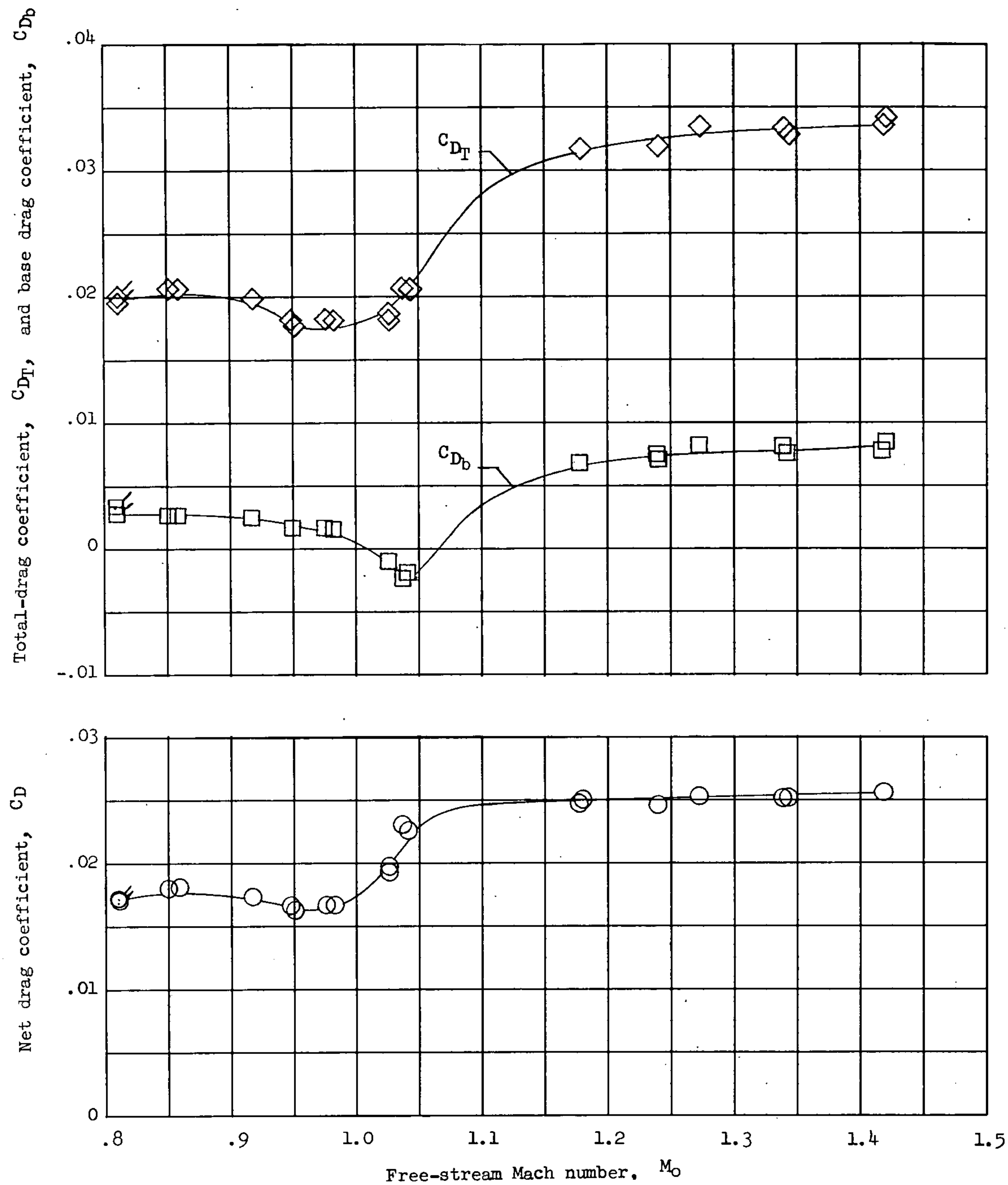
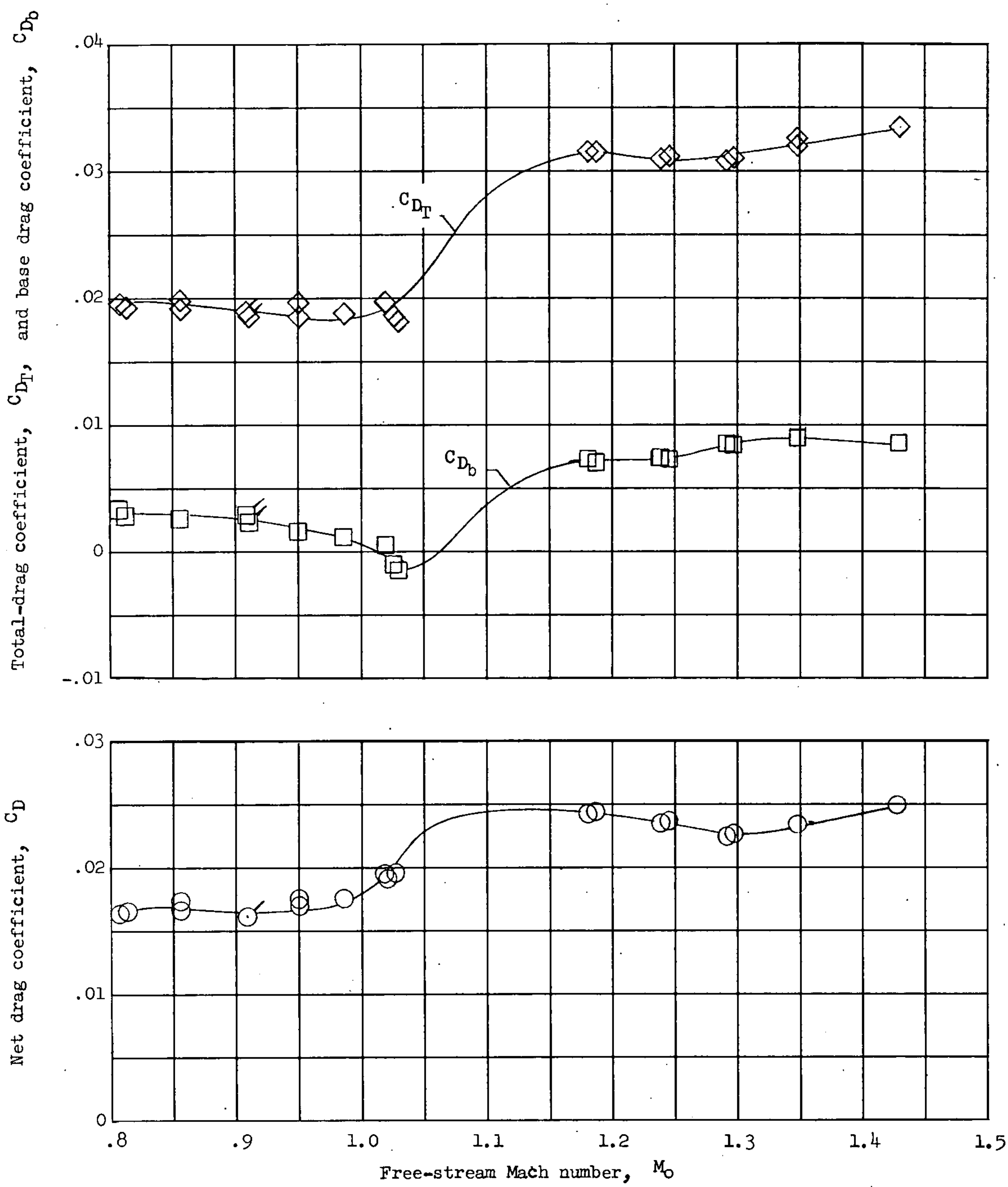


Figure 3.- Longitudinal area developments of the configurations tested.



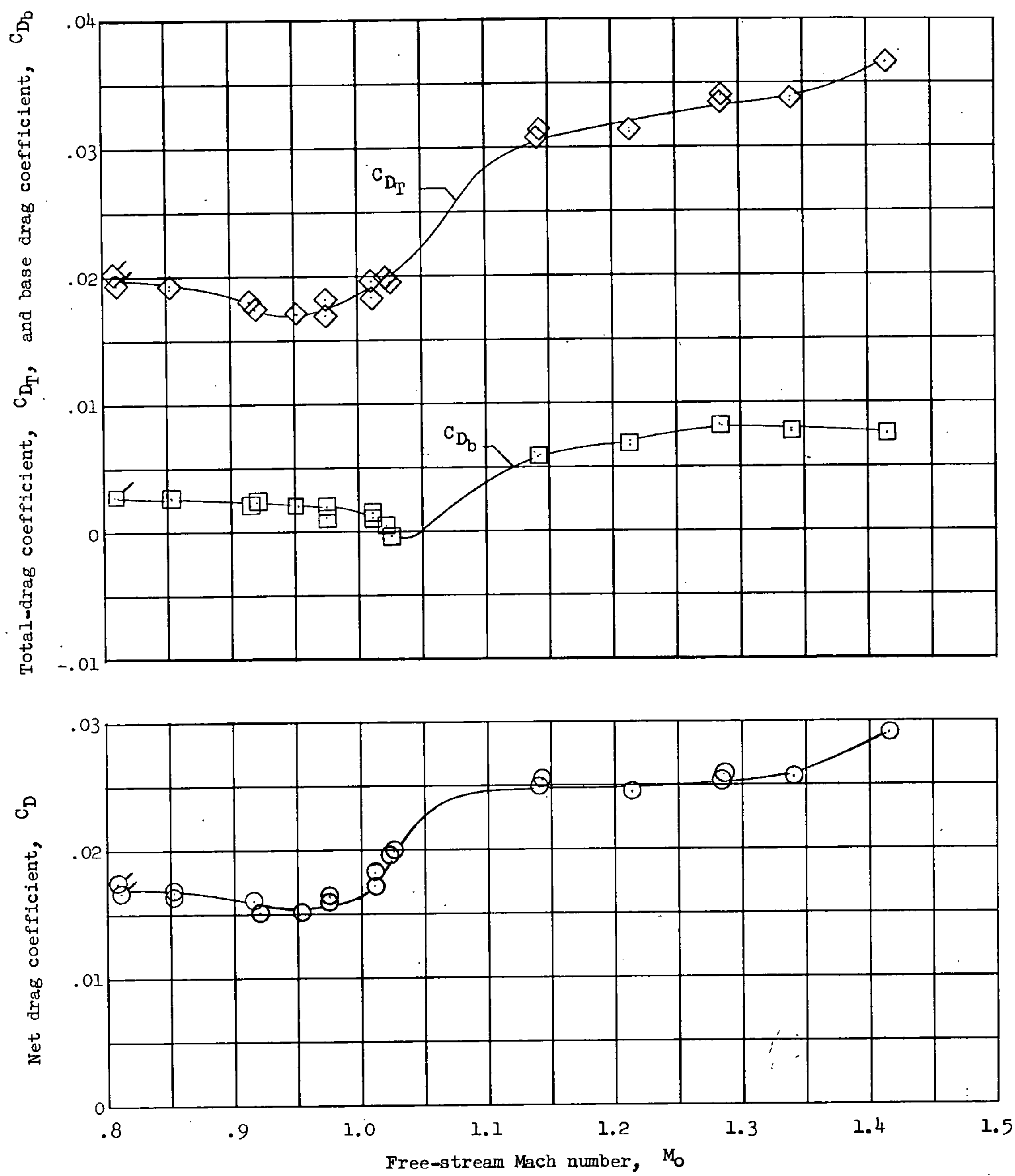
(a) Fuselage indented according to the supersonic area rule.

Figure 4.- The variation of total-drag coefficient, base drag coefficient, and net drag coefficient with Mach number for the three configurations tested. $\alpha = 0^{\circ}$.



(b) Combination fuselage.

Figure 4.- Continued.



(c) Fuselage indented according to the transonic area rule.

Figure 4.- Concluded.

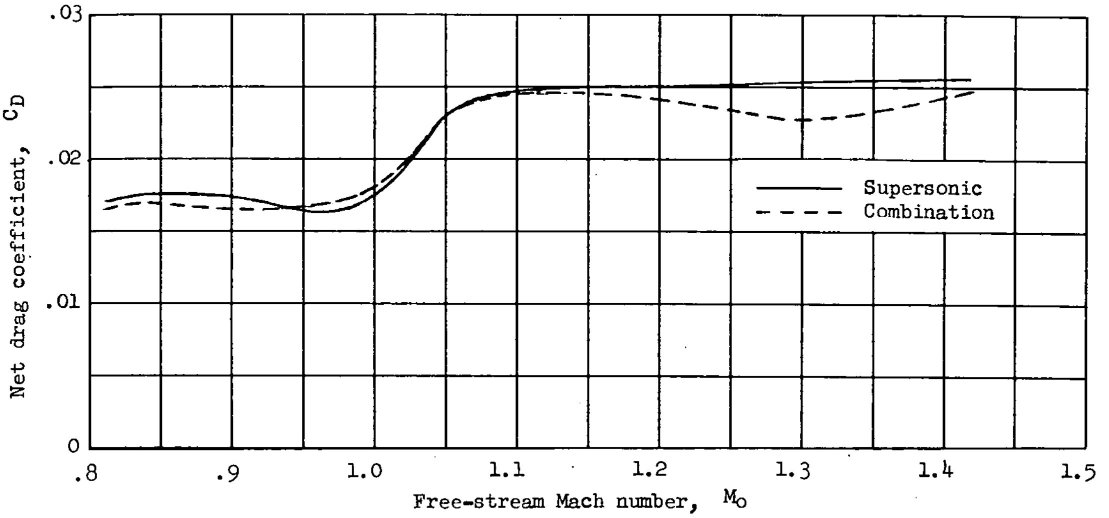


Figure 5.- A comparison of the variation of net drag coefficient with Mach number for the combination fuselage configuration and fuselage indented according to the supersonic area rule.

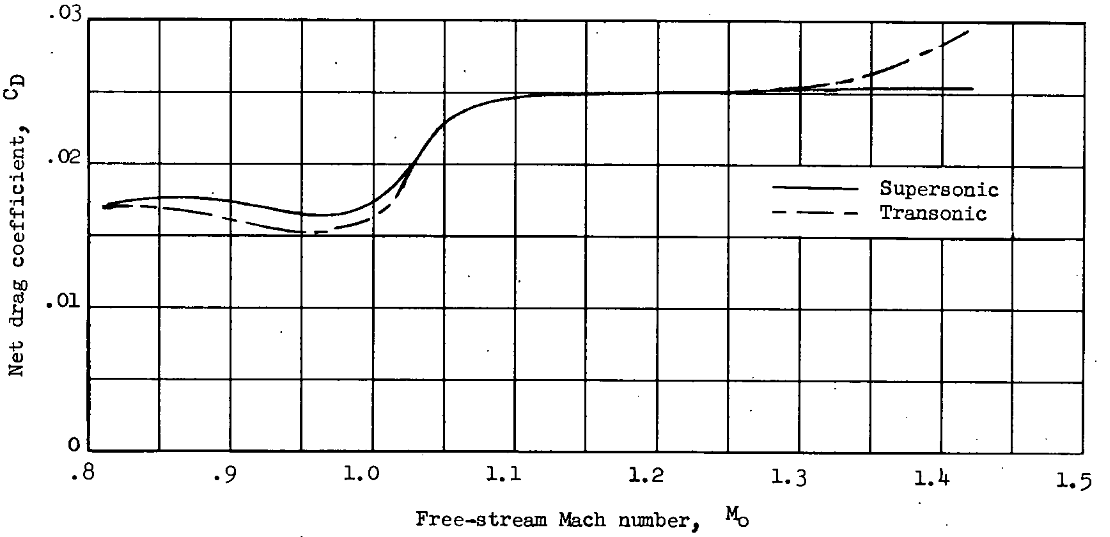


Figure 6.- A comparison of the variation of net drag coefficient with Mach number for the fuselage configurations indented according to the supersonic and transonic area rules.

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